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## CONVERGING PIN COOLED AIRFOIL

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3 [0001] The U.S. Government may have certain rights in this invention pursuant to contract  
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## BACKGROUND OF THE INVENTION

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8 [0002] The present invention relates generally to gas turbine engines, and, more specifically,  
9 to turbine blade cooling therein.

10 [0003] In a gas turbine engine, air is pressurized in a multistage compressor and mixed with  
11 fuel for generating hot combustion gases in a combustor. The gases are discharged through a  
12 high pressure turbine (HPT) which powers the compressor, typically followed by a low  
13 pressure turbine (LPT) which provides output power by typically powering a fan at the  
14 upstream end of the engine. This turbofan configuration is used for powering commercial or  
15 military aircraft.

16 [0004] Engine performance or efficiency may be increased by increasing the maximum  
17 allowed operating temperature of the combustion gases that are discharged to the HPT which  
18 extracts energy therefrom. Furthermore, engines are continually being developed for  
19 increasing cruise duration and distance, for one exemplary commercial application for a  
20 supersonic business jet and for an exemplary military application such as a long range strike  
21 aircraft.

22 [0005] Increasing turbine inlet temperature and cruise duration correspondingly increases  
23 the cooling requirements for the hot engine components, such as the high pressure turbine  
24 rotor blades. The first stage rotor blades receive the hottest combustion gases from the  
25 combustor and are presently manufactured with state-of-the-art superalloy materials having  
26 enhanced strength and durability at elevated temperature. These blades may be configured  
27 from a myriad of different cooling features for differently cooling the various portions of the  
28 blades against the corresponding differences in heat loads thereto during operation.

29 [0006] The presently known cooling configurations for first stage turbine blades presently  
30 limit the maximum allowed turbine inlet temperature for obtaining a suitable useful life of the

1 blades. Correspondingly, the superalloy blades are typically manufactured as directionally  
2 solidified materials or monocrystal materials for maximizing the strength and life capability  
3 thereof under the hostile hot temperature environment in the gas turbine engine.

4 [0007] The intricate cooling configurations found in the blades are typically manufactured  
5 using common casting techniques in which one or more ceramic cores are utilized. The  
6 complexity of the cooling circuits in the rotor blades is limited by the ability of conventional  
7 casting processes in order to achieve suitable yield in blade casting for maintaining  
8 competitive costs.

9 [0008] Like the first stage turbine blades, the first stage turbine nozzle includes hollow vanes  
10 which require suitable cooling for extended life while exposed to the hot combustion gases.  
11 The vanes, like the blades have corresponding airfoil configurations, and include internal  
12 cooling circuits of various configurations specifically tailored to cool the different parts of the  
13 vanes corresponding with the different heat loads from the combustion gases.

14 [0009] Accordingly, it is desired to provide a turbine airfoil having an improved cooling  
15 configuration for further advancing temperature and durability thereof in a gas turbine engine.

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#### 17 BRIEF DESCRIPTION OF THE INVENTION

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19 [0010] A turbine airfoil includes pressure and suction sidewalls extending in chord between  
20 leading and trailing edges and in span between a root and a tip. A septum is spaced between  
21 the sidewalls to define two cooling circuits on opposite sides of the septum which converge  
22 between the leading and trailing edges. An array of pins extends inwardly from the pressure  
23 sidewall at a discharge end of the circuits, and the pins decrease in length to conform with the  
24 converging circuit.

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#### 26 BRIEF DESCRIPTION OF THE DRAWINGS

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28 [0011] The invention, in accordance with preferred and exemplary embodiments, together  
29 with further objects and advantages thereof, is more particularly described in the following  
30 detailed description taken in conjunction with the accompanying drawings in which:

1 [0012] Figure 1 is an axial sectional view in elevation of an exemplary high pressure turbine  
2 rotor blade having pin bank sidewall cooling.

3 [0013] Figure 2 is a radial sectional view of the blade airfoil illustrated in Figure 1 and taken  
4 along line 2-2.

5 [0014] Figure 3 is an enlarged isometric view of a portion of the airfoil shown in Figure 2  
6 illustrating one embodiment of the pin bank configuration disposed in the pressure sidewall  
7 upstream from the trailing edge.

8 [0015] Figure 4 is an isometric view of another portion of the airfoil shown in Figure 2  
9 illustrating a second embodiment of the pin bank located along the trailing edge.

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## 11 DETAILED DESCRIPTION OF THE INVENTION

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13 [0016] Illustrated in Figure 1 is an exemplary first stage turbine rotor blade 10 for use in a  
14 gas turbine engine in a high pressure turbine immediately downstream from a combustor  
15 thereof. The blade may be used in an aircraft gas turbine engine configuration, or may also be  
16 used in non-aircraft derivatives thereof.

17 [0017] The blade includes a hollow airfoil 12 extending radially in span outwardly from a  
18 supporting dovetail 14 joined together at a common platform 16. The dovetail may have any  
19 conventional configuration including dovetail lobes or tangs which mount the blade into a  
20 corresponding dovetail slot in the perimeter of a turbine rotor disk (not shown). The dovetail  
21 is joined to the integral platform by a shank therebetween.

22 [0018] The airfoil 12 includes a concave pressure sidewall 18 and a laterally or  
23 circumferentially opposite convex suction sidewall 20. The two sidewalls are joined together  
24 at axially or chordally opposite leading and trailing edges 22,24, and are spaced apart  
25 therebetween. The airfoil sidewalls and edges extend radially in span from an inner root 26 to  
26 an outer tip 28. The dovetail is integrally joined to the airfoil at the platform disposed at the  
27 airfoil root which defines the radially inner boundary for the combustion gases which flow  
28 around the airfoil during operation.

29 [0019] As shown in Figures 1 and 2, the airfoil further includes an imperforate wall or  
30 septum 30 spaced generally in the middle between the pressure and suction sidewalls 18,20 to

1 define with those sidewalls two independent cooling circuits 32,34 on opposite sides of the  
2 septum.

3 [0020] The septum 30 commences behind the leading edge 22 integrally with the suction  
4 sidewall 20, and terminates in front of the trailing edge 24 integrally with the pressure  
5 sidewall 18 for generally splitting in two the airfoil along the camber line. Disposed  
6 immediately behind the leading edge 22 is a third cooling circuit 36.

7 [0021] The three circuits 32,34,36 are preferably independent from each other, and each  
8 receives cooling air 38 through a corresponding inlet extending radially through the dovetail  
9 14 and platform 16. The cooling air is typically compressor discharge air suitably channeled  
10 from the compressor (not shown) of the gas turbine engine.

11 [0022] The several cooling circuits are specifically configured or tailored for suitably  
12 cooling their respective portions of the airfoil for withstanding the corresponding heat loads  
13 from combustion gases 40 which flow over the external surfaces of the airfoil during  
14 operation.

15 [0023] The third cooling circuit 36 may have any conventional configuration, and may  
16 include corresponding bridges extending between the pressure and suction sidewalls of the  
17 airfoil which define two corresponding flow channels 36 that extend radially in span behind  
18 the leading edge of the airfoil and between the opposite sidewalls. The center bridge between  
19 the two channels includes a row of impingement holes through which a portion of the cooling  
20 air 38 is firstly directed in impingement against the internal surface of the airfoil behind the  
21 leading edge. The leading edge includes several rows of film cooling holes 42 which then  
22 discharge the spent impingement air along the external surfaces of the airfoil for providing  
23 film cooling in a conventional manner.

24 [0024] However, the two side-cooling circuits 32,34 extend in span along opposite sides of  
25 the middle septum 30 for providing enhanced cooperation therebetween and enhanced cooling  
26 therefrom. In particular, the first circuit 32 is disposed along the inside of the pressure  
27 sidewall 18, and the second circuit 34 is disposed along the inside of the suction sidewall 20,  
28 and extends in part aft from the first circuit to the trailing edge 24.

29 [0025] As best illustrated in Figure 2, the two cooling circuits 32,34 both converge between  
30 the leading and trailing edges in the axial or chordal direction for correspondingly accelerating

1 the cooling air therethrough during operation.

2 [0026] Correspondingly, a bank or array of turbulator pins 44,46 extends transversely  
3 inwardly from the pressure sidewall 18 at the discharge end of one or both of the two circuits  
4 for providing local mesh cooling in their coverage region or area. The pins bridge the  
5 converging circuits and correspondingly decrease in length to conform with the converging  
6 profiles of the circuits in the axial or chordal direction between the leading and trailing edges.

7 [0027] The two circuits 32,34 include corresponding outlets spaced chordally apart on the  
8 pressure sidewall for discharging the cooling air from the respective circuits. The first circuit  
9 32 includes a first outlet in the form of a radially elongate slot 48 extending in span along the  
10 pressure sidewall. The second circuit includes a second outlet in the form of a radial row of  
11 outlet apertures 50 which extend axially between the pressure and suction sidewalls and  
12 terminate adjacent the trailing edge 24.

13 [0028] As shown in Figures 2 and 3, the first array of pins 44 is disposed in the first circuit  
14 32 immediately upstream or forward from the first outlet slot 48 for discharging the cooling  
15 air 38 in a continuous film along the span of the slot.

16 [0029] The first circuit 32 preferably consists of a single channel extending in radial span  
17 along the pressure sidewall to provide a common inlet converging to the array of first pins 44,  
18 which channel continues to converge in the axially aft direction to the common outlet slot 48.  
19 In this way, the cooling air is initially channeled radially upwardly through the dovetail into  
20 the first circuit channel 32 and then is distributed along the full height of the bank of first pins  
21 44, which redirect the cooling air axially aft towards the common outlet slot 48. If desired,  
22 one or more radially aligned outlet slots 48 may be used.

23 [0030] This configuration provides many advantages. Firstly, the bank of first pins 44 are  
24 preferably spaced apart both in span and chord along the pressure sidewall 18 for providing a  
25 circuitous flowpath immediately behind the pressure sidewall for providing enhanced cooling  
26 thereof, with the spent cooling air then being discharged through the common outlet slot 48  
27 for providing a continuous film of cooling air downstream therefrom to the airfoil trailing  
28 edge 24. The local mesh cooling effected by the bank of pins 44 provides enhanced cooling in  
29 this local region of the pressure sidewall which is subject to high heat loads from the external  
30 combustion gases that flow thereover during operation.

1 [0031] The axially converging first circuit 32 accelerates cooling air therethrough and  
2 between the first bank of pins 44, with the spent cooling air then being diffused in the  
3 common outlet slot 48 prior to discharge over the pressure sidewall. The turbulator pins at the  
4 forward or inlet side of the pin bank are correspondingly longer than those at the aft or outlet  
5 end of the pin bank and correspondingly generate more turbulence in the cooling air. The  
6 longer pins also have more heat transfer area for enhancing heat transfer from the hot pressure  
7 sidewall.

8 [0032] Correspondingly, the shorter pins near the outlet of the first circuit may be used to  
9 limit the flow area between the pins and meter or control the flowrate of the cooling air  
10 discharged through the first circuit. Collectively, the first pins 44 of short to long length  
11 provide heat conduction between the hot pressure sidewall and the relatively cold internal  
12 septum 30 which splits the airfoil in two parts.

13 [0033] The septum 30 itself is cooled on both surfaces thereof by the corresponding first and  
14 second cooling circuits 32,34 and provides an improved heat sink for the heat conducted  
15 through the first pin bank 44. Since the septum 30 splits the airfoil in two parts, each part,  
16 including the corresponding circuits 32,34, has a relatively large width, which increases the  
17 strength of the corresponding ceramic cores which may be used in the casting process for the  
18 manufacture of the airfoils using conventional practice. Thicker cores are preferred over  
19 thinner cores to increase the strength thereof, and correspondingly increase the effective yield.  
20 Thin cores are problematic and increase difficulty of casting, and typically result in smaller  
21 yields.

22 [0034] As shown in Figure 2, the second circuit 34 preferably includes a plurality of  
23 imperforate, transverse bridges 52 which integrally join together the suction sidewall 20 and  
24 the septum 30 to define a three-pass serpentine circuit which discharges the cooling air  
25 through the second outlet aperture 50 at the airfoil trailing edge. The first pass or channel of  
26 the second circuit includes an inlet extending through the dovetail, and shown in Figure 1, and  
27 the three channels converge in the axially aft direction illustrated in Figure 2 as the suction  
28 sidewall and septum converge together toward the trailing edge.

29 [0035] As illustrated in Figure 2, one of the cold bridges 52 in the second circuit 34  
30 integrally joins together the suction sidewall 20 and the septum 30 directly behind the first pin

1 array 44. This bridge provides additional conduction for removing heat from the first pin  
2 array 44. This bridge also increases the stiffness of the airfoil between the pressure and  
3 suction sidewalls in the location of the first pin array 44.

4 [0036] Accordingly, the hot pressure sidewall 18 illustrated in Figure 2 is cooled by the  
5 cooperation of the single channel first circuit 32 in which the cooling air directly cools the first  
6 bank of pins 44 by internal convection and conduction, followed in turn by using the spent  
7 cooling air to form a cooling film discharge from the outlet slot 48.

8 [0037] The heat input from the combustion gases flowing over the suction sidewall 20 is  
9 typically less than that from the pressure sidewall, and the three-pass serpentine second circuit  
10 34 may be used for channeling another portion of the cooling air independently from the first  
11 circuit, and cooling the suction sidewall in turn along the corresponding portions of the  
12 serpentine circuit. The spent serpentine cooling air is then discharged through the last channel  
13 of the converging second circuit 34 through the decreasing-size bank of second pins 46 for  
14 discharge through the trailing edge 24.

15 [0038] The trailing edge outlet apertures 50 illustrated in Figure 2 are positioned in the  
16 middle between the pressure and suction sidewalls near the root of the airfoil illustrated in  
17 Figure 1. However, the outlet apertures 50 may breach the pressure sidewall of the airfoil  
18 immediately short of the trailing edge as the trailing edge decreases in thickness along the  
19 span of the airfoil as also illustrated in Figure 1.

20 [0039] As shown in Figures 1, 2, and 4, the second array of turbulator pins 46 may be used  
21 at the discharge end of the second circuit 34 in combination with the first array of turbulator  
22 pins 44 disposed in the discharge end of the first circuit 32. The bank of second pins 46  
23 illustrated in Figures 2 and 4 is located immediately upstream from the row of second outlet  
24 apertures 50, and correspondingly decreases in length as the two sides of the airfoil converge  
25 together to the trailing edge.

26 [0040] In this configuration, the second pin array 46 is disposed downstream from the first  
27 pin array 44 immediately aft of the outlet slot 48, and integrally joins together the pressure and  
28 suction sidewalls in the trailing edge region of the airfoil.

29 [0041] Like the first pin array 44, the second pin array 46 provides enhanced cooling of the  
30 pressure sidewall due to the decreasing length of the turbulator pins therein, and the

1 converging portion of the second circuit flow channel 34. However, the cold septum 30  
2 terminates before the bank of second pins 46, and therefore does not provide the additional  
3 cooling advantage found with the first pin array 44.

4 [0042] The corresponding turbulator pins 44,46 of the two mesh arrays are similarly spaced  
5 apart both in span and chord along the pressure sidewall for providing corresponding  
6 circuitous flowpaths for discharging cooling air from the airfoil.

7 [0043] The pins 44,46 in the two banks may have uniform spacing as illustrated in Figures 3  
8 and 4, or may have variable spacing as the specific design permits. The pins 44,46 may have  
9 any suitable configuration such as uniform configurations being generally square for the first  
10 pins 44 and being generally cylindrical for the second pins 46. The pins may be staggered as  
11 illustrated, or may be disposed in line from row to row.

12 [0044] The banks of turbulator pins disclosed above cooperate with the converging cooling  
13 circuits for providing enhanced local cooling of the airfoil along the pressure sidewall which  
14 typically receives maximum heat load from the hot combustion gases during operation. The  
15 mesh pins may be used with various forms of the cooling circuits, and with other conventional  
16 features for providing tailored cooling of the different regions of the airfoil. The cooling  
17 circuits may be varied in configuration, and additional internal straight turbulators may also be  
18 used in the various cooling channels. The pressure and suction sidewalls may include various  
19 rows of the film cooling holes as required for enhancing the cooling thereof in conventional  
20 manners.

21 [0045] Accordingly, the combination of mesh cooling and conventional cooling features  
22 permits the designer more flexibility in defining the specific features of the cooling  
23 configuration of the airfoil for minimizing the use of cooling air therein, while maximizing the  
24 local cooling performance of the limited air. Although the cooling configurations disclosed  
25 above are found in a gas turbine engine high pressure turbine rotor blade, the mesh cooling  
26 may also be provided in turbine nozzle vanes for corresponding cooling enhancement.

27 [0046] While there have been described herein what are considered to be preferred and  
28 exemplary embodiments of the present invention, other modifications of the invention shall be  
29 apparent to those skilled in the art from the teachings herein, and it is, therefore, desired to be  
30 secured in the appended claims all such modifications as fall within the true spirit and scope of

- 1 the invention.
- 2 [0047] Accordingly, what is desired to be secured by Letters Patent of the United States is
- 3 the invention as defined and differentiated in the following claims in which we claim: